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COVER SHEET FOR TECHNICAL MEMORANDUM

TITLE- Small Space Propulsion Stage Preliminary Design - PM-II

TM-69-1012-1

DATE- January 16, 1969

FILING CASE NO(S)- 730

AUTHOR(S)- A. E. Marks

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ABSTRACT

The use of a small propulsion stage for attitude control, midcourse velocity corrections, and high velocity missions with small payloads is investigated. The commonality aspect of a single stage design for lunar, planetary, and earth orbital missions is shown to be feasible.

The mission applications are: (1) the direct return of crew and payload from the lunar surface, (2) orbit-keeping and attitude control for an earth orbital mission, (3) midcourse corrections and attitude control for planetary missions, and (4) planetary abort.

Four propellant combinations, LF_2/LH_2, LO_2/LH_2, $FLOX/CH_4$, and Compound A/MHF-5, representing high energy cryogenics, space storable, and earth storable propellants, are studied. The stage gross weights varied between 26,000 and 32,000 pounds. The two most promising candidate stages use LF_2/LH_2 and $FLOX/CH_4$ propellants. These are close in gross weight and are about 2000 pounds lighter than a Compound A/MHF-5 stage and 4000 pounds lighter than LO_2/LH_2.

Detailed analyses of the stage designs are performed and the procedures and results contained in a series of Appendices.

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SUBJECT: Small Space Propulsion Stage Preliminary Design - PM-II Case 730

DATE: January 16, 1969

FROM: A. E. Marks

TM-69-1012-1

TECHNICAL MEMORANDUM

1.0 INTRODUCTION

Manned space missions require the use of both large and small propulsion stages. Large stages provide velocity increments for such maneuvers as out-of-orbit injections, deceleration, etc. Small propulsion stages are mainly used for attitude control and midcourse corrections, and can also be used for high velocity requirements with small payloads. This report summarizes the design aspects of such a small space propulsion stage; the propulsion module II (PM-II).

The PM-II was originally conceived as a planetary mission propulsion system to perform interplanetary midcourse corrections, attitude control, station-keeping in planetary orbits and post-injection abort. Consistent with a policy of seeking hardware commonality[22] across many mission areas, attention is addressed to the applicability of the PM-II to other mission areas, lunar and earth orbit.

Three classes of propellants were evaluated--cryogenics, space-storables, and earth-storables--to see if one lends itself most easily to the variety of missions.

The possible uses of the PM-II and estimated mission requirements are shown in Table 1. Analysis of these missions leads to the sizing of the PM-II for post injection planetary abort. Applicability of that design to other mission areas is then evaluated. A detailed description of the preliminary stage sizing is presented in Appendix A.

TABLE 1

PM-II-MISSIONS

Mission	PM-II Function	Propulsion Reqmt.	Payload	Mission Duration
Lunar	Direct Return of Crew and Payload from Lunar Surface.	ΔV = 10,000 fps	14,200 lbs	Up to 2 years (dormant) on lunar surface.
Earth Orbit	Orbit-Keeping and attitude control.	$I_T \approx 10^7$ lb-sec	50,000 to 200,000 lbs	2 years
Planetary	Midcourse corrections and attitude control.	ΔV = 500 fps/leg I_T = 1000 lb-sec/day	100,000 to 200,000 lbs	2 years plus 180 days in earth orbit
Planetary	Post-Injection Abort	ΔV = 9000 fps	19,200 lbs	10 days

2.0 SUMMARY OF RESULTS

PM-II commonality is quite feasible for lunar, planetary and earth orbital missions. The stages sized for planetary abort are about the desired size for the lunar and planetary flyby missions and also for earth orbital missions in the 100 to 200 n. mi. altitude range. The propellants resulting in the best system performance were LF_2/LH_2 and $FLOX/CH_4$. The LO_2/LH_2 stage does not appear competitive for these applications. The difference in performance between all the stages does not justify omitting any from future studies, since more intangible comparisons (cost, handling problems, toxicity, launch implications, etc.) may significantly effect final propellant selection.

A summary of the stage weights for the various missions and the propellant mass fractions (λ')are shown in Table 2. It is assumed that the tanks are filled to 85 percent capacity (15 percent ullage) for all missions. Boiloff will vary with the missions and significantly effect the λ' that can be obtained.

TABLE 2

PM-II DESIGN SUMMARY

	LF_2/LH_2	LO_2/LH_2	$FLOX/CH_4$	Compound A/MHF-5
*Gross Weight (Earth Launch)	27,741	31,988	26,462	28,544
λ'	.845	.827	.895	.909
#Weight (Planetary Abort)	26,630	29,841	25,819	28,104
λ'(Planetary Abort)	.889	.877	.919	.923
Weight (after 2 years on lunar Surface)	22,631	25,873	24,685	28,544
λ'(Lunar)	.808	.785	.890	.910

*Stage loaded to accomplish lunar mission taking boiloff into account

#Meteoroid shielding jettisonned

3.0 MISSION DESCRIPTIONS

3.1 Planetary Abort

The operational features of the planetary abort function are complex. During injection from earth orbit onto the planetary trajectory, the Earth Entry Module (EEM) must be docked to the PM-II at the aft end of the EEM. This configuration allows the most efficient and fastest means of initiating abort. The abort package is first separated from the rest of the spacecraft. If PM-II meteoroid shielding is then jettisoned*, the resulting abort stage configuration propellant fraction, λ', is increased up to 6.0 points. The proper spacecraft alignment is then established before main engine firing.

Abort ΔV requirements increase very rapidly with time after injection. MSC states that it could take up to 5 minutes after trans-planetary injection to determine (from the trajectory) if abort is necessary. This requirement will size the engine thrust level, since for engines of the 10,000 to 20,000 pound thrust class, the PM-II engine run time can approach 10 minutes. The abort ΔV requirements are based on instantaneous impulse, and a much more detailed analysis of the abort problem using a finite burn time is necessary to adequately determine these requirements. The stage was sized with the abort accomplished in 10 minutes after injeciton. The stage was also sized for a residual injection velocity, V_∞, of 0.21 emos to encompass most of the contemplated Mars and Venus flyby missions.

3.2 Planetary Missions

The planetary mission selected for this design study was a dual planet flyby. This mission has a nominal 740 day duration. It is assumed that there is a 180 day earth orbit stay time before embarking on the planetary phase. The three legs of this mission are 160 days to Venus encounter, 280 days from Venus to Mars, and a 300 day return to Earth from Mars. The PM-II must provide attitude control continuously for the entire mission, and 500 fps midcourse ΔV for each of the three legs of the mission. The use of CMG's for attitude stabilization might eliminate the need for propulsive attitude control but was not considered so as to be conservative in the propulsion requirements. It is assumed that the spacecraft weighs 200,000 pounds and leaves 50,000 pounds of probes and expendables during each planetary encounter. The resulting earth return spacecraft would then weigh about 100,000 pounds.

---

*The practicality of shield jettisoning has not been investigated.

3.3 Lunar Mission

The lunar mission profile consists of a standard Saturn V launch and a PM-I[8,9] lunar injection and landing with the PM-II as part of the landed lunar payload. A dormant period of up to 2 years on the lunar surface is assumed. The PM-II will then be activated and provide the lunar ascent and return ΔV for a total of over 14,200 pounds which includes an EEM and the Environmental Control System and Life Support System stage addition.*

3.4 Earth Orbital Missions

The types of earth orbital missions contemplated for the PM-II are low altitude (100 n.mi.) earth resources type missions, and solar and stellar astronomy missions in higher orbits (250 n.mi.). The low altitude missions are flown with the spacecraft in a belly-down orientation, while the higher altitude missions have the spacecraft quasi-inertially oriented. The spacecraft physical properties, mass, length, diameter, etc., are estimated from a study on multi-disciplinary space stations[23]. The requirements for these two year missions and the PM-II capability are shown in Figure 1. It can be seen that the PM-II is quite applicable to lower altitude missions, but grossly oversized at higher altitudes.

3.5 Constraints

Before the design study is presented, the constraints upon which the design is based should be understood. From an earth orbit logistics standpoint, the PM-II should be compatible in diameter with Gemini or Apollo earth entry modules and the Titan III or SIB launch vehicle families. This dictates a diameter of about 10 to 15 feet.

The standard Saturn V earth launch vehicle (ELV) plus PM-I delivered lunar payload capability presents another limitation on the PM-II design since about 45,000 pounds can be delivered to the lunar surface. This includes the EEM, the PM-II, and lunar payload. Therefore, the maximum PM-II gross weight at earth launch should be about 30,000 pounds or less, allowing about 15,000 lb for the EEM.

The planetary mission EEM size used during this analysis is 17,500 pounds (based on a four-man crew with an earth entry velocity of 55,000 fps). For the lunar return mission, the EEM weighs 12,500 pounds (four men and 36,000 fps entry speed). Neither EEM is capable of independently sustaining life for more than about

* Appendix H

24 hours, and a lifetime of up to two weeks may be required. A small stage section is added to the EEM's to provide the essential life support and environmental control systems. This stage addition will weigh about 1700 pounds.

Larger crew sizes were not considered at this time since the resulting lunar EEM would be too heavy to allow the mission to be accomplished with the standard Saturn V. A larger crew would also increase the planetary PM-II and EEM weights.

4.0 DESIGN CONSIDERATIONS

4.1 Stage Mass Fraction

An extensive analysis of meteoroid protection, thermal environment, stage structure and packaging was necessary to allow reasonable estimations of the stage inert weights. The details of these fractions attainable for the various propellant combinations are shown in Table 3 with major component weights delineated.

The requirements of the lunar mission (two years dormant on the lunar surface) present the most severe thermal problems with maximum boiloff occurring. To insure that enough propellant remains to complete the mission, additional propellant must be loaded at earth launch to compensate for the boiloff. The result is a relatively heavy stage at earth launch--26,000 to 32,000 pounds depending on propellant selection.

To minimize design changes between missions, the propellant tanks necessary for the lunar mission are used for all other contemplated missions and assumed fully loaded (15 percent ullage). For planetary abort, almost no boiloff will have occurred and all the loaded propellant is usable. Less than lunar boiloff will occur during planetary and earth orbital missions and the remaining boiloff contingency propellant will also be usable. It should be noted, however, that more fuel than oxidizer boiloff occurs and the resulting tanked mixture ratio is quite low. The engines must therefore be designed to run very fuel rich to utilize all the propellants and this results in reduced specific impulse. The total impulse capability of the stages is still increased even at the reduced specific impulse. This puts a stringent requirement on engine development, however, and other schemes of propellant utilization may be employed. Also, the stages need not be loaded to full propellant capacity for some of the missions.

The capability of these stages for the various missions is shown in Table 4. With the boiloff propellant added to that determined by the planetary abort mission, the specific impulse for the cryogenic stages drops almost 15 percent if the engine is run at these reduced (tanked) mixture ratios, but the total impulse capability is still increased. The FLOX/CH_4 mixture ratio is only slightly lowered with a negligible effect on engine specific impulse. Since no boiloff of Compound A/MHF-5 propellants occurs, there will be no change in the tanked mixture ratio.

TABLE 3

WEIGHT SUMMARY AND MASS FRACTION

GROSS WEIGHT AT EARTH LAUNCH

	LO_2/LH_2	LF_2/LH_2	$FLOX/CH_4$	Comp A/MHF-5
Loaded Propellant (Boiloff Contingency)	27,086 (6,115)	23,849 (5,110)	23,917 (1,777)	26,121 (0)
Meteoroid Shield and Structure	1,555	1,015	793	687
Insulation	129	98	77	32
Storability Penalty (Tanks and Meteoroid Shielding)	1,446	1,130	30	--
Structure and Subsystems	1,772	1,649	1,645	1,704
Total Weight	31,988	27,741	26,462	28,544
λ' (assuming boiloff propellant contingency usable)	.827	.845	.895	.910
PLANETARY ABORT				
ΔW Meteoroid Shield Jettison	-2147	-1381	-643	-440
Total Weight	29,841	26,360	25,819	28,104
λ' (using all loaded propellant)	.885	.890	.918	.922
LUNAR ASCENT AND RETURN (After two years dormant on lunar surface)				
ΔW Propellant Boiloff	-6115	-5110	-1777	0
Total Weight	25,873	22,631	24,685	28,544
λ'	.785	.808	.890	.910

(Table 3 Cont.)

PLANETARY FLYBY

	LO_2/LH_2	LF_2/LH_2	$FLOX/CH_4$	Comp A/MHF-5
ΔW Propellant Boiloff	-3850	-2680	0	0
Total Weight	28,138	25,061	26,462	28,544
λ′	.802	.826	.895	.910

EARTH ORBITAL

	LO_2/LH_2	LF_2/LH_2	$FLOX/CH_4$	Comp A/MHF-5
ΔW Propellant Boiloff	-4170	-2880	0	0
Total Weight	27,818	24,861	26,462	28,544
λ′	.800	.825	.895	.910

TABLE 4

PM-II CAPABILITY

	LO_2/LH_2	LF_2/LH_2	$FLOX/CH_4$	Comp A/MHF-5
Tanked Mixture Ratio, O/F	2.15	3.53	5.62	2.70
Design Mixture Ratio, O/F	6.00	12.00	5.75	2.70
Engine Specific Impulse, sec. (Tanked M.R.)	410	400	400	350
Engine Specific Impulse, sec. (Design M.R.)	460	470	400	350
Planetary Abort ΔV, fps	10,200	9,400	9,600	9,000
Planetary ΔV/leg (3 legs), fps	627	576	610	590
Lunar Return Payload lbs.(including EEM)	15,400	15,100	16,000	15,600
Earth Orbit Total Impulse, lb-sec	9.3×10^6	8.5×10^6	9.5×10^6	9.1×10^6

On planetary missions, if post-injection abort is necessitated, the additional propellant loaded for boiloff contingency is utilized and results in an increase in the abort ΔV of 400 fps for LF_2/LH_2, 600 fps for $FLOX/CH_4$, and 1200 fps for LO_2/LH_2. If abort is not initiated and the planetary missions are flown, the ΔV capability for each leg of the mission is increased over 20 percent to around 600 fps.

The lunar ascent and return mission payloads that can be returned by the PM-II are 15,000 to 16,000 pounds. Included in this payload is the EEM and the LSS adapter section totaling about 14,200 pounds.

The total impulse capability of the PM-II for a two year earth orbital mission varies from 8.5 to 9.5 million lb-sec. From Figure 1, the applicability for this stage size for two year missions is in the 100 to 200 n.mi. orbital range.

4.2 Propulsion Module Configuration

The PM-II stage design was analyzed for one, two, and three engine configurations. The pros and cons of these designs are discussed in detail in Appendix F. Figures 2, 3, and 4 are schematics of these designs.

5.0 CONCLUSIONS

The major conclusion to draw from this study is that the PM-II has a great deal of versatility and can be used for a wide variety of missions. Lunar, planetary, and some earth orbital missions all require about the same size propulsion stage and commonality of design is clearly indicated. A stage gross weight of 26,000 to 32,000 pounds is needed to satisfy these mission requirements.

Evaluation of high performance cryogenic, space-storable and earth-storable propellants does not yet indicate one propellant to be superior. On a performance basis, LF_2/LH_2 and $FLOX/CH_4$ are better than LO_2/LH_2 and Compound A/MHF-5. The key technology area to aid in propellant selection is the storability problem. The boiloff problem for the cryogenics and in some cases the space-storables imposes severe weight penalties on the stages. Mass fractions attainable with cryogenics are less than 0.85, space-storables are near 0.90 and earth-storables over 0.90.

Considerable detailed design and analysis is required to more accurately define the PM-II. Some of the areas of most concern requiring further investigation are delineated below:

1. Thermal Analysis: A detailed analysis of the thermal environment during transplanetary, lunar and Earth orbital missions is needed. The effect of the thermal cycling from lunar days and nights must also be evaluated.

2. Fluorine Environment: The effect of fluorine venting and engine exhaust products on sensors, optical equipment, etc., and also on space suits for extra-vehicular activity should be determined.

3. Refrigerators and Subcooling: An investigation of subcooling techniques and problems is necessary, and preliminary design of refrigeration system is also necessary if it is determined that such a system is required.

4. Meteoroid Shielding: An accurate determination of the meteoroid environment and shielding requirements is necessary since this weight is significant. The shielding should be designed such that launch loads are also handled by this structure.

5. Abort Requirements: A finite burntime analysis of the planetary abort maneuver is required to size the stage adequately. The time necessary to accomplish the preliminary preparations prior to abort must also be established.

6. Subsystems: Detailed design of all stage subsystems is required for accurate stage definition. The subsystem weights now presented are mainly prorated from other stage studies.

It is recommended that a PM-II type stage be considered by NASA as a candidate future spacecraft propulsion stage. It has versatility and can be used with large and small spacecraft for a wide variety of missions. The technology evolved from such a program would be extremely valuable to both manned and unmanned missions.

Acknowledgements

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A. E. Marks

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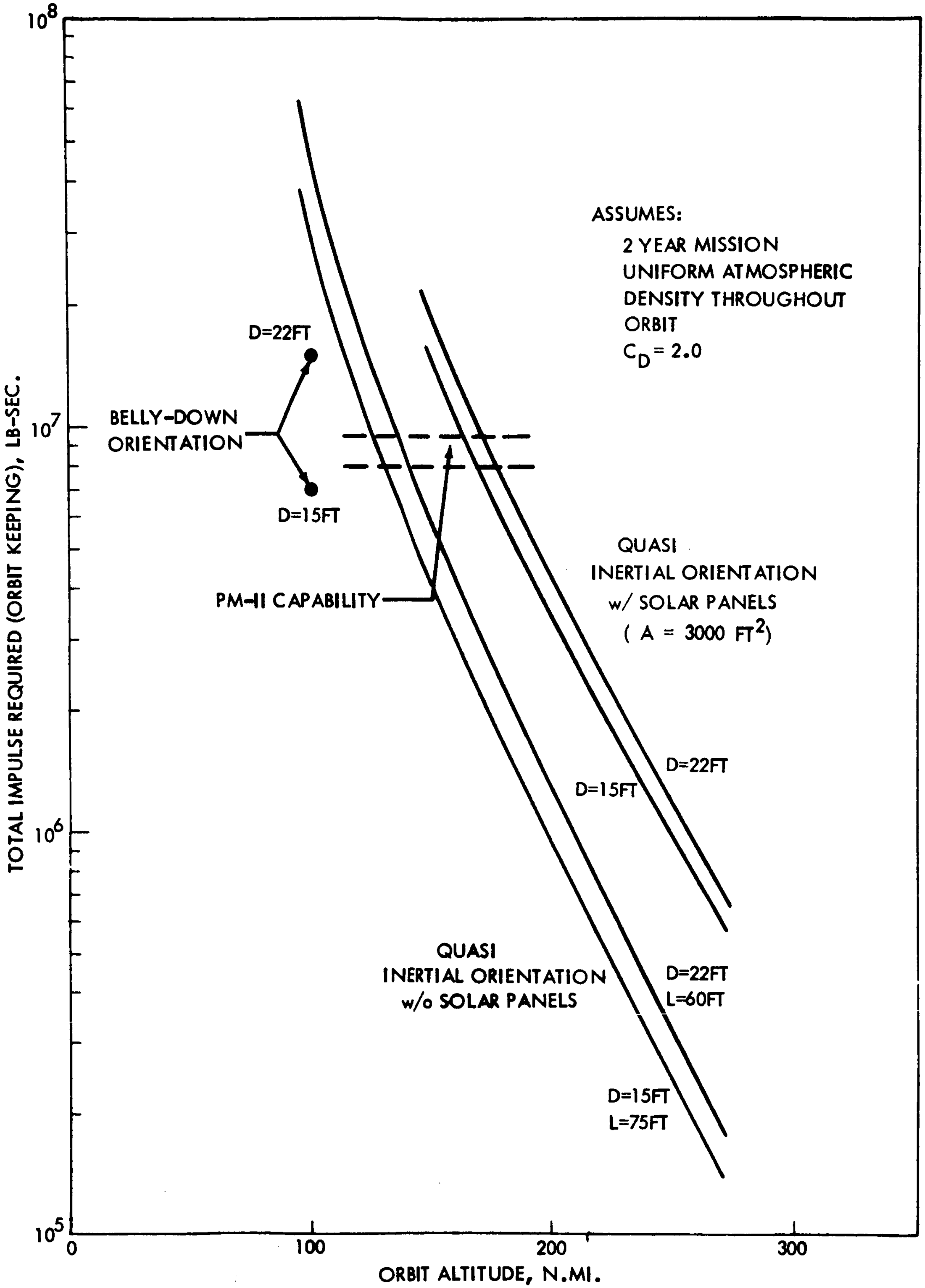

FIGURE 1 - ORBIT - KEEPING REQUIREMENTS

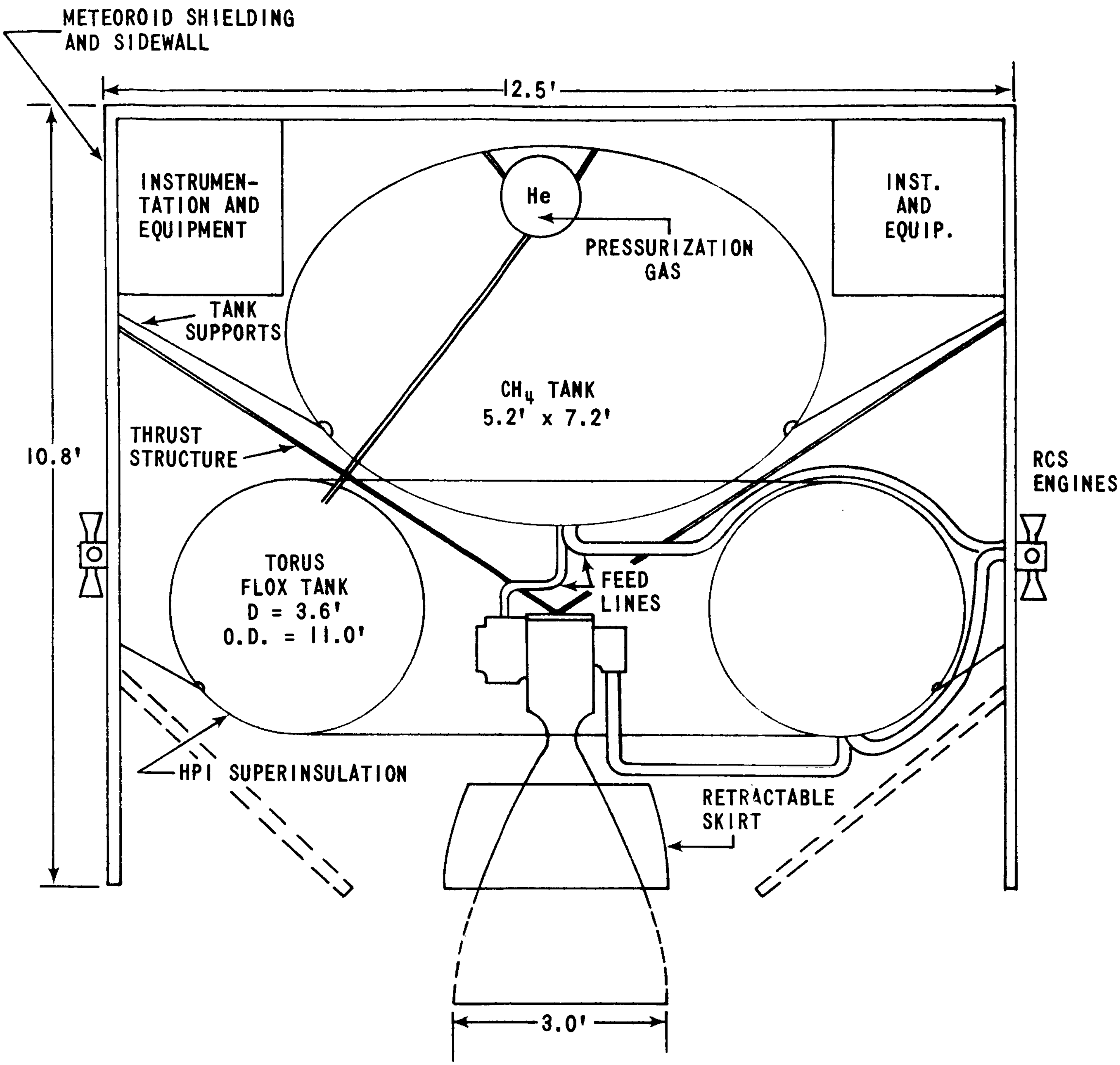

FIGURE 2 - PROPULSION MODULE II FLOX/CH_4 I ENGINE

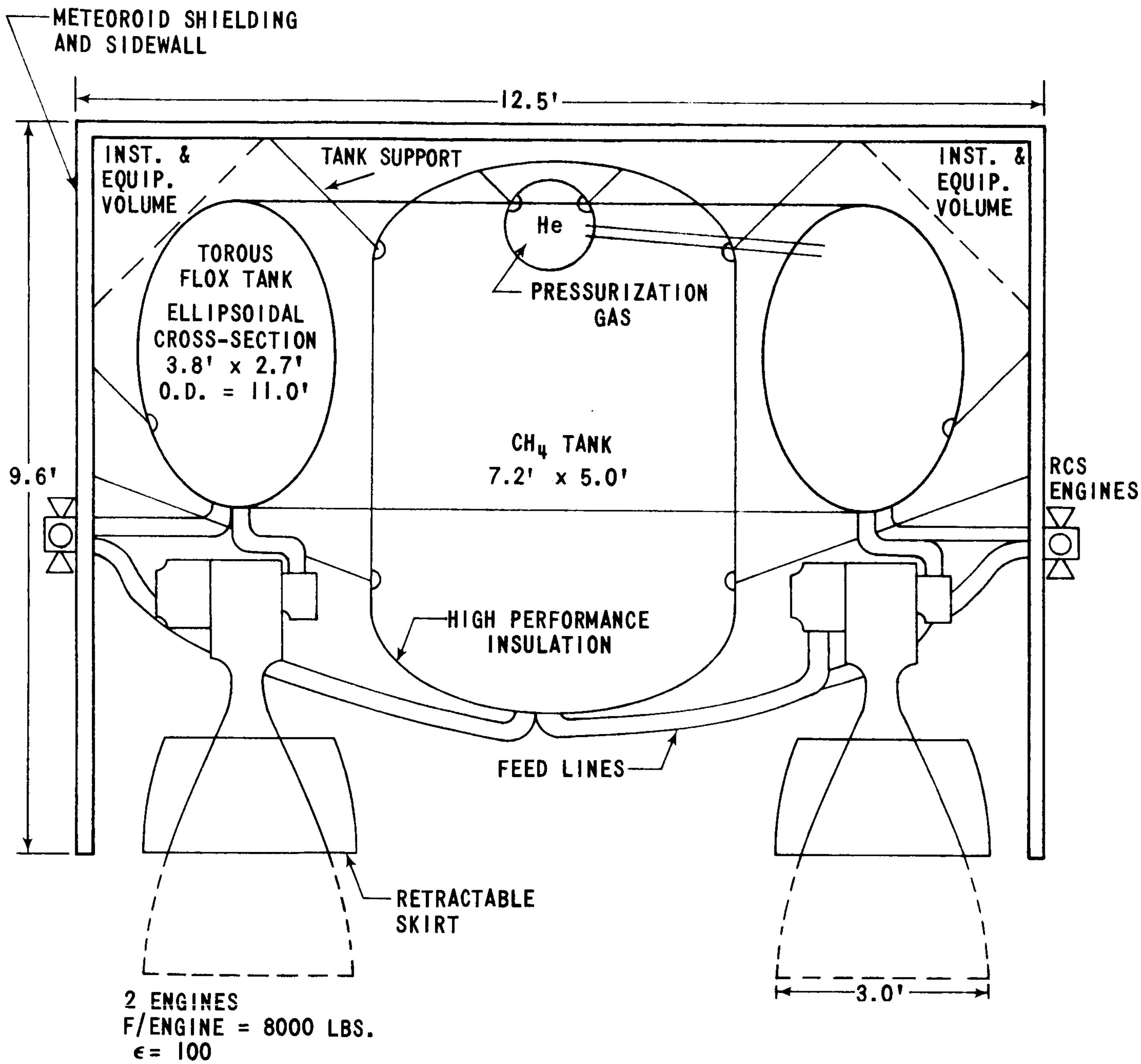

FIGURE 3 - PROPULSION MODULE II FLOX/CH_4 2 ENGINES

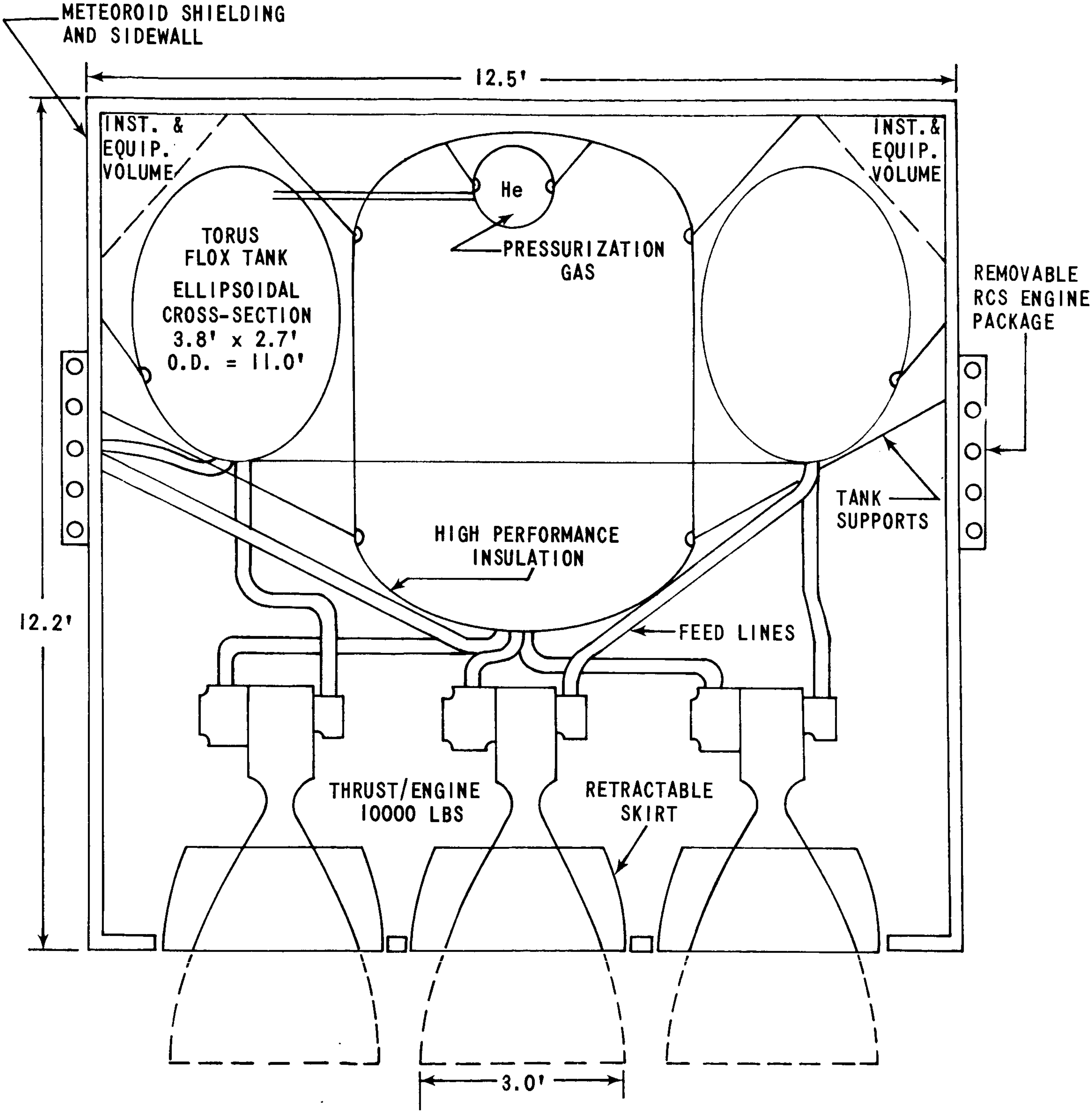

FIGURE 4 - PROPULSION MODULE II FLOX/CH_4 3 ENGINES

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APPENDIX A

PRELIMINARY SIZING SUMMARY

Planetary-Midcourse Corrections and ACS

A midcourse correction of 500 fps ideal velocity increment is assumed necessary for each leg of the planetary mission. It is also assumed that the ACS requirements are 1000 lb.-sec/day total impulse enroute.

A range of specific impulse values were studied (360, 400, 435, 460 seconds) with stage mass fractions of 0.80, 0.85, and 0.90. This allows a parametric look at the sizing problem.

The mission profile is a dual planet flyby with a total duration of about 740 days. Enroute time to Venus is 160 days, Venus to Mars is 280 days, and 300 days return to earth. The spacecraft weight is assumed to be 200,000 pounds with 50,000 pounds ejected or expended at each planet. Midcourse corrections are needed enroute to Venus, between Venus and Mars, and during the Earth return leg.

Since the total impulse of a rocket engine is the weight of propellant times the specific impulse, the weight of propellant for attitude control is easily found: $W_P = I_T/I_{sp}$. The weight of propellant necessary for the midcourse corrections is found from the ideal velocity equation. An iterative solution is made for the midcourse corrections on each leg to determine initial propellant loading and hence stage gross weight. The tabulated results for the various mission functions are listed in Table A1.

TABLE A1

PLANETARY MISSION PROPELLANT REQUIREMENTS, LBS.

Attitude Control	I_s = 360 sec.	400	435	460
Outbound	445	400	368	348
Venus	778	700	644	609
Return	834	750	690	653

Appendix A (Cont'd)

Midcourse Corrections	I_s = 360	400	435	460
Outbound				
λ' = 0.80	11400	9800	8650	7920
.85	10860	9320	8230	7620
.90	10500	9000	8050	7400
Venus to Mars				
λ' = .80	8300	7100	6230	5690
.85	7900	6750	5920	5470
.90	7620	6500	5780	5300
Return				
λ' = .80	5210	4400	3820	3460
.85	4930	4160	3610	3310
.90	4750	4000	3520	3200
Total Propellant Requirements				
λ' = .80	25350	21700	19020	17420
.85	24500	20930	18600	17010
.90	23700	20250	18040	16550
Stage Gross Weight				
λ' = .80	31700	27100	23800	21800
.85	28800	24600	21900	20000
.90	26400	22500	20000	18400

Planetary - Post Injection Abort

The velocity requirements for initiating abort shortly after injection onto a planetary orbit were determined from North American Aviation[1] and McDonnell-Douglas Company[2] sources. Figure A1 shows the abort ΔV requirements as a function of injection residual velocity, V_∞, and time of abort after injection. The return time is three days. Figure A2 shows the variation of the abort velocity requirements with return time. It should be noted that the abort ΔV reduction with increasing trip return time is very small after about three days.

Appendix A (cont'd)

The maximum energy mission considered was one having a V_∞ of 0.21 emos. This is inclusive of most of the Mars and Venus flyby missions presently contemplated. The abort time is considered at 10 minutes after injection to include abort decision time, abort initiation, earth entry module (EEM) and PM-II separation from spacecraft, and engine run time. Based on these assumptions, an abort ΔV requirement of 9000 fps is required.

It is assumed that a 17,500 pound earth entry module (EEM) and a small stage addition (~1700) carrying the necessary life support (LSS) and environmental control (ECS) equipment to sustain life up to ten days is also returned. The stage size is then:

	I_s = 360 sec	400	453	460
λ' = .80	39,900 lbs	32,500	28,000	25,400
.85	33,500	27,900	24,300	22,100
.90	28,800	24,400	21,400	19,700

Lunar Ascent and Return

The lunar ascent and return mission requires a ΔV of about 10,000 fps. Assuming an EEM of 12,500 pounds, and the LSS and ECS equipment used for planetary abort, the required PM-II stage size is found to be:

	I_s = 360 sec	400	435	460
λ' = .80	37,000 lbs	29,000	25,300	22,400
.85	30,300	24,700	21,300	19,300
.90	25,500	21,100	18,600	17,000

The planetary abort function is seen to require the largest size stage. Using this stage, its application to the other missions is evaluated and summarized in Table A2. It should be noted that these stage sizes are based on usable propellant, only, and no boiloff allowances are made.

TABLE A2

PM-II SIZING - PLANETARY ABORT

FUNCTION	PARAMETER		I_s = 360	400	435	460	COMMENTS
Planetary Abort	PM-II Gross W., lbs.	λ'=.80	39,900 lbs.	32,500	28,000	25,400	V_∞=0.21 emos, t_{abort}=10 min.
		.85	33,500	27,900	24,300	22,100	t_{return}=3 to 10 days,
		.90	28,800	24,400	21,400	19,700	W_{return}=19,200 lbs. ΔV = 9,000 fps.
Lunar	P.L. delivered to lunar surface (excluding PM-II and EEM), lbs.	.80	------ lbs.	500	5,000	7,600	Delivered W_G=45,000 lbs.
		.85	------	5,100	8,700	10,900	$W_G=W_{EEM}+W_{PM-II}$+PL
		.90	4,200	8,600	11,600	13,300	W_{EEM}=12,500 + 1,720 lbs.
	P.L. returned to Earth (excluding EEM), lbs.	.80	------ lbs.	1,330	1,670	1,760	ΔV return = 10,000 fps.
		.85	-------	1,370	2,000	1,900	$W_{return} = W_{EEM}$ + PL
		.90	1,770	2,030	2,100	2,200	
E.O.	Total I/Day, lb-sec.	.80	19,700 lb-sec	17,800	16,700	16,400	2 years in orbit
		.85	16,500	15,300	14,500	13,900	
		.90	14,200	13,400	12,800	12,400	
Planetary	Surplus Prop. (Excess over Mis. requirements), lbs.	.80	6,500 lbs.	4,900	2,800	1,400	I_T = 1,000 lb. -sec/day
		.85	4,100	2,400	1,560	700	ΔV midcourse = 500 fps/leg
		.90	3,000	1,100	900	300	300 $W_{G\ S/C}$ = 200,000 lbs.

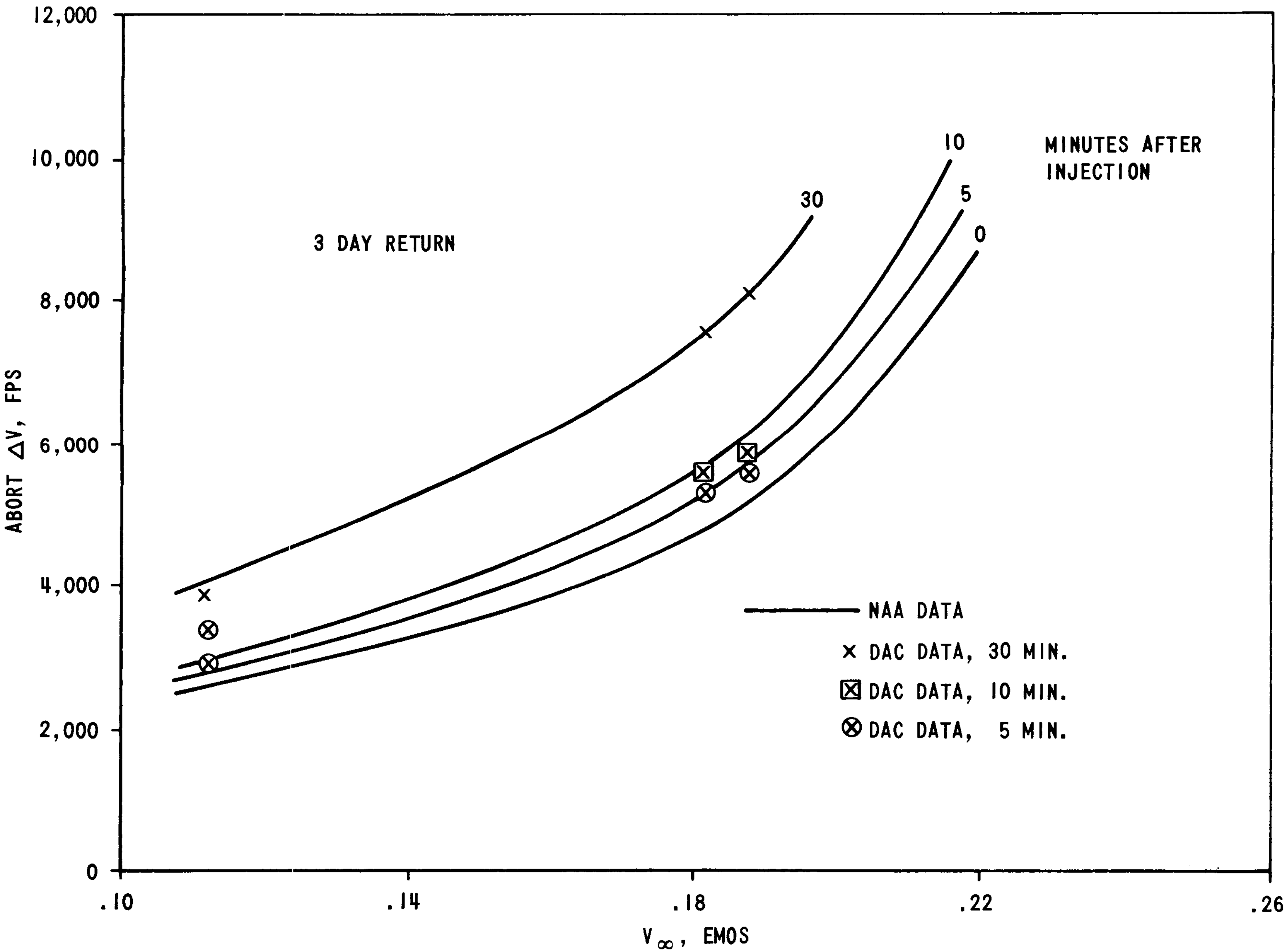

FIGURE A1 - PLANETARY ABORT REQUIREMENTS

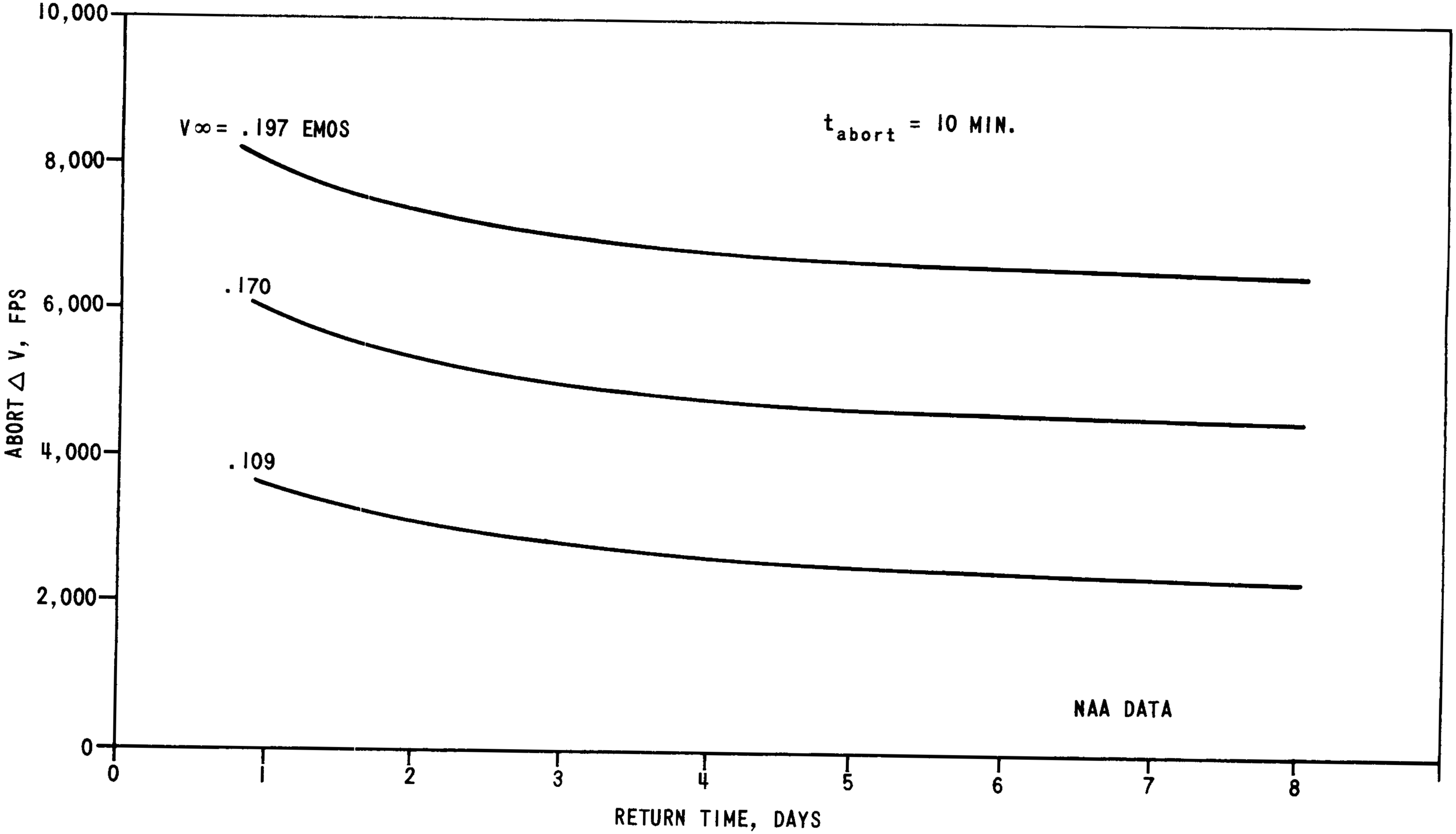

FIGURE A2 - PLANETARY ABORT REQUIREMENTS

APPENDIX B

TANK DESIGNS

The amount of propellant necessary for the PM-II is determined from the planetary abort requirements using the ideal velocity equation.

$$\Delta V_i = I_s \; g \; \ln R$$

where

ΔV_i = ideal velocity, fps

I_s = vacuum specific impulse, sec.

$R = \frac{W_o}{W_f}$ = initial gross weight/final weight after propellant burn

Also, $W_o = W_I + W_P + W_{PL}$

$W_f = W_I + W_{PL}$

$\lambda' = W_P/(W_I + W_P)$

where

W_I = stage inert, weight, lbs.

W_P = stage propellant weight, lbs.

W_{PL} = payload weight, lbs.

λ' = propellant mass fraction

Appendix B (Cont'd)

Then

$$R = \frac{\frac{1}{\lambda'} W_p + W_{PL}}{\frac{1-\lambda'}{\lambda'} W_P + W_{PL}}$$

Knowing the necessary ΔV, the engine specific impulse, and the payload, and estimating the propellant mass fraction, the propellant weight is determined:

$$R = e^{\frac{\Delta Vi}{I_s g}}$$

$$W_P = \frac{(R-1)\ \lambda'}{(1-R+R\lambda')} W_{PL}$$

The volume occupied by these propellants is then determined once the propellant mixture ratio is assumed.

$$MR = \frac{W_{ox}}{W_f}$$

where W_{ox} = oxidizer weight, lbs.

W_f = fuel weight, lbs.

Hence

$$V_f = \frac{W_f}{\rho_f} \frac{1}{(MR+1)}$$

$$V_{ox} = \frac{W_{ox}}{\rho_{ox}} = \frac{W_P}{\rho_{ox}} \frac{MR}{(MR+1)}$$

Appendix B (Con'd)

where

V = propellant volume, ft^3

ρ = propellant density, lb/ft^3

Assuming there is a 15 percent ullage volume, the required tank volumes are $V_T = 1.15\ V_P$. With the tank volumes known, the tank dimensions are easily found for the various configurations by:

Type	Volume	Surface Area	Comments
Sphere	$4/3\pi r^3$	$4\pi r^2$	r = radius
$\sqrt{2}$ Ellipsoid	$4/3\pi a^2 b$	$2\pi a^2+\pi\frac{b^2}{e}\ln\frac{1+e}{1+e}$	a = major semi-axis b = minor semi-axis a/b = eccentricity = $\sqrt{2}$
Toroid	$2\pi r^2 R$	$4\pi^2 rR$	r = cross-section radius R = toroid radius to center of cross-section

Table B1 presents a propellant tank dimensions summary.

TABLE B1

PROPELLANT TANK SUMMARY

	LO_2	LH_2	LF_2	LH_2	FLOX	CH_4	Comp.A	MHF-5
Volume, ft^3	280	775	212	385	241	142	183	125
Torus tanks								
Diameter, ft	11.0	---	11.0	---	11.0	---	11.0	---
Height, ft	3.8	---	3.0	---	3.6	---	2.9	---
Surface area, ft^2	270	---	237	---	263	---	206	---
$\sqrt{2}$ Ellipsoidal tanks								
Diameter, ft	9.1	12.8	8.3	10.2	8.7	7.3	7.8	6.9
Height, ft	6.5	9.1	5.9	7.2	6.1	5.2	5.5	4.9
Surface Area, ft^2	227	448	189	280	205	144	166	129
Spherical Tanks								
Diameter, ft	8.1	11.4	7.4	9.0	7.7	6.5	7.0	6.1
Surface area, ft^2	206	409	172	255	188	132	152	177

Appendix B (Cont.)

	LO_2	LH_2	LF_2	LH_2	FLOX	CH_4	Comp. A	MHF-5
Stage Length w/o Engines, ft								
Torus-Ellipse		---		9.5		7.5		6.5
Torus-Sphere		14.6		11.0		8.8		7.0
Ellipse-Ellipse		---		13.5		11.8		10.9

Propellant Tank Weights

A spherical container stressed by uniform internal pressure has a required thickness

$$t = \frac{PR}{2\sigma}$$

where P = Internal pressure
R = Tank radius
σ = Allowable stress

The weight of the container is

$$W = \frac{APR}{2\sigma} \rho$$

For a sphere, $V = \frac{R}{3}A_s$, so $W = \frac{3}{2} \frac{\rho}{\sigma} PV$.

For an ellipse, the thickness is

$$t = \frac{P}{2\sigma}(a^2 + 2y^2)^{1/2}$$

a = semimajor axis

y = distance measure along minor axis.

For a constant stress shell, the weight of an ellopsoid is found to be

$$W_e = \int PT \, dA = a\frac{P\rho}{\sigma} V_e.$$

For torus tanks, the required thickness is

$$t = \frac{Pr}{2\sigma}$$

where

r = cross-sectional radius

Appendix B (Cont'd)

The weight of the tank is then found to be

$$W = 4\pi^2 rRt\rho = \frac{2\pi}{r} t\rho V = \pi \frac{\rho}{\sigma} PV.$$

The tank weights for the various designs, using aluminum, are then:

TABLE B2

Propellant Tank Weights, lbs

	LO_2	LH_2	LF_2	LH_2	FLOX	CH_4	Comp. A	MHF-5
Sphere	118	384	117	191	119	83	26	18
Torus	247	805	245	400	249	174	55	38
$\sqrt{2}$ Ellipse	157	511	156	254	159	111	35	24

The tank design pressures are calculated and discussed in Appendix D.

APPENDIX C

METEOROID SHIELDING AND SIDEWALL STRUCTURE

For the long duration missions contemplated in this study, the meteoroid shielding necessary to assure a mission reliability of .999 becomes a significant part of the stage inert weight. The weight of the shielding per square foot of surface area also exceeds that needed for the structural loads encountered during launch and thrusting. Based on the data from Boeing,[10] Figure C1 was estimated and shows the required wall weights per square foot of surface area for a given compressive launch load and for mission durations of 2 and 5 years. From a Bellcomm source,[11] the required single sheet aluminum thickness is estimated as $t_{AL} = 131 \times 10^{-4}\ (AT)^{1/3}$, where A is the exposed surface area of the tanks in ft^2 and T is time in days. This formula includes a spall factor of 1.5. The required sheet thickness for all the stages is on the order of 1 inch for 2 year missions and 1.25 inches for 5 year missions. These thicknesses are impractical and the use of multi-sheet walls are employed. With proper spacing and design, it is reasonable to assume a "bumper factor" of 7.5 for the outer wall.

The side wall compressive launch loads are determined to be a maximum of 600 pounds per running inch of circumference. This is based on a weight above the PM-II at launch of just over 37,000 pounds and a maximum acceleration of 7.5g's. This includes the launch escape system, and about 1700 pounds for the extended LSS and ECS stage for the EEM.

The meteoroid shielding and sidewall weight for 2 and 5 years missions, with a 600 lb in running load are estimated from Boeing data to be 2.1 and 3.2 lb/ft^2 respectively. In this region, the structure loads dominate the shield thickness and the effects of vehicle area are minimized.

The meteoroid and sidewall requirements for the candidate stages are summarized below. It can be seen that the results from both methods of calculation are fairly close. It is also noted that these weights were determined under the assumption that the asteroid belt would not be encountered.

Appendix C (Cont'd)

TABLE C1

METEOROID SHIELDING AND SIDEWALL REQUIREMENTS

	W/A, lb/ft^2 ($t = 131 \times 10^{-4}(AT)^{1/3}$)		Boeing Data	
	2 years	5 years	2 years	5 years
LO_2/LH_2	2.25	3.08	2.1	3.2
LF_2/LH_2	2.07	2.76	2.1	3.2
$FLOX/CH_4$	1.92	2.68	2.1	3.2
Compound A/MHF-5	1.84	2.53	2.1	3.2

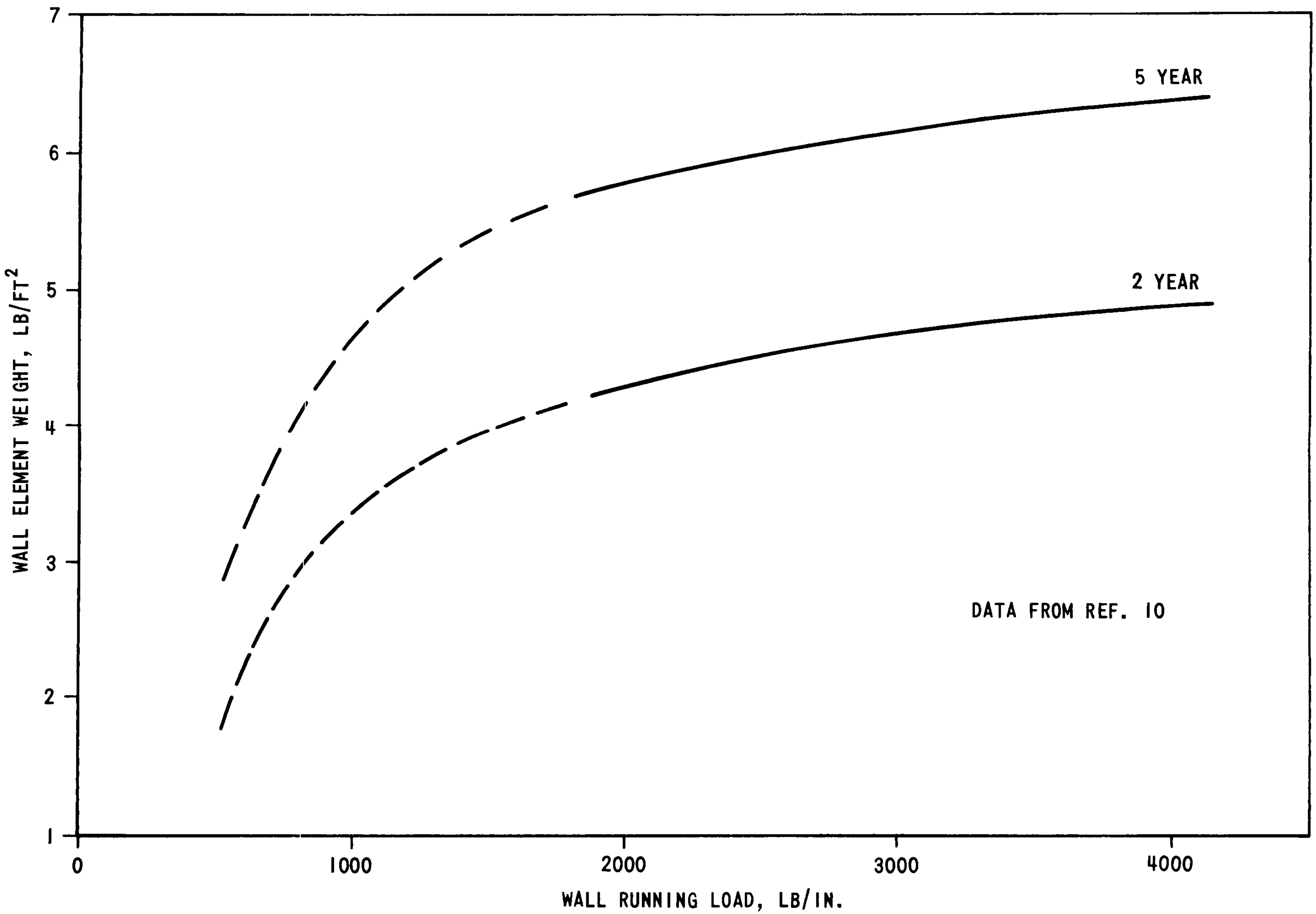

FIGURE C1 - METEOROID SHIELDING REQUIREMENTS

APPENDIX D

THERMAL ANALYSIS

A critical problem in a vehicle design of this type is the long term storage of the cryogenic propellants. To estimate the magnitude of this problem, a preliminary thermal analysis was performed by assuming outside tank wall temperatures (outside surface of insulation). These temperatures, which were assumed uniform over the entire tank surfaces, were 300°R for planetary missions, 400°R for earth orbital missions, and 500°R on the lunar surface. These assumptions are fairly consistent with current studies within the industry[24] and are somewhat conservative. The lunar surface temperatures will actually be cyclic with the two week lunar days and nights and the assumption of a 500°R constant temperature should be quite conservative. Obviously, these are gross assumptions and a more detailed analysis of the thermal environment is necessary. It was also assumed that heat shorts contributed one third of the total heat input. In all cases, this assumption resulted in a higher heat transfer rate than an empirical formula developed by the Martin Co.[13] predicts:

$$q_{H.S.} = \frac{\left(\frac{D}{10}\right)^{2.36} \left(\frac{T_H}{100}\right)^{1.67} \rho_p^{0.5}}{\left(\frac{T_c}{10}\right)^{0.8}}$$

where the tank is assumed spherical,

D = diameter, ft
T_H = hot wall temperature, °R
ρ_p = propellant density, lb/ft^3
T_c = cold wall or propellant temperature, °R
$q_{H.S.}$ = heat flux due to heat shorts, BTU/hr

Appendix D (cont'd)

The largest heat transfer rate through shorts (one third of the total heat input) was used to be conservative and also to account for the non-spherical tanks. NRC-2 super-insulation was used (1.5 inches) with a conductivity of 3×10^{-5} BTU/ft^2 -hr-°R/ft. All the propellants and vapor are considered in equilibrium (saturated vapor).

The total heat transfer rate into the propellants is:

$$\dot{q}_{total} = \dot{q}_{ins} + \dot{q}_{H.S.}$$

where

$\dot{q}_{ins}$ = heat transfer rate through the insulation, BTU/hr

But, it is assumed $\dot{q}_{H.S.} = 1/3\ \dot{q}_{total}$, so that $\dot{q}_{total} = 1.5\ \dot{q}_{ins}$

The heat transfer rate through the insulation is $\dot{q}_{ins} = \frac{kA\Delta T}{t}$

where k = insulation conductivity = 3×10^{-5} BTU/ft^2 -hr-°R/ft
A = tank surface area, ft^2
ΔT = temperature drop across insulation, °R
t = insulation thickness, ft

The temperature drop across the insulation under the assumed outer boundary temperature is:

TABLE D1

	ΔT,°R			
	LH_2	LO_2	LF_2,FLOX	CH_4
Earth Orbit	350°R	250	250	200
Planetary	250	150	150	100
Lunar	450	350	350	300

Appendix D (cont'd)

The resulting insulation heat flux, $\dot{q}_{ins}/A$, in BTU/ft^2 -day is:

TABLE D2

	Earth Orbit	Planetary	Lunar
LH_2	2.01 BTU/ft^2-day	1.44	2.59
LO_2	1.44	.86	2.01
LF_2,FLOX	1.44	.86	2.01
CH_4	1.15	.58	1.72

Using the torus oxidizer tank designs and the spherical and $\sqrt{2}$ ellipsoidal fuel tanks, and dividing by the propellant mass, the total heat rate per pound of propellant can be found:

$$\dot{q}_{total}/m = 1.5 \quad \dot{q}_{ins}/A \times \frac{A_{tank}}{m_{prop.}} \quad \text{BTU/lb -day}$$

these rates are then:

TABLE D3

$\dot{q}/m$

	Earth Orbit	Planetary	Lunar
LO_2 (torus)	.033 BTU/lb-day	.019	.046
LH_2 (Sphere)	.426	.306	.549
LF_2 (torus)	.030	.018	.042
LH_2 (ellipse)	.586	.420	.757
(sphere)	.534	.384	.688
FLOX (torus)	.030	.018	.042
CH_4 (ellipse)	.076	.039	.114
(sphere)	.070	.036	.105

(Appendix D (cont'd)

From IBM calculations done at Douglas Aircraft Co.[14] the tank pressure versus total heat input per pound of propellant is determined knowing the initial pressure and percent ullage volume. These results are shown in figure D1. The total heat input for a two year lunar mission, a two year planetary (assuming 180 days in earth orbit) and a two year earth orbital mission are easily found since the heat rates are known. These total heat inputs are:

TABLE D4

Q/m, BTU/lb.

	Earth Orbit	Planetary	Lunar
LO_2	24	20	34
LH_2	311	300	401
LF_2	22	18.5	31
LH_2	418	394	522
FLOX	22	18.5	31
CH_4	54	35	80

It would be desirable to fly all the missions without venting the propellant tanks. A zero-g vent system is a difficult design problem and larger propellant boiloff will occur with venting. The lunar mission, which has the worst heating problem, will determine the tank design conditions and the heat inputs are so high that all the propellant tanks must be vented. The time the tanks could remain non-vented assuming 100 psia maximum tank pressure buildup (68 psia for LH_2) is found knowing the heating rates and the allowable heat per pound.

$$t_{w/o\ venting} = \frac{(Q/m)_{allow}}{(\dot{q}/m)_{total}}$$

TABLE D5

Days Without Venting

	Earth Orbit	Planetary	Lunar
LO_2	479 days	832	343
LH_2	68	95	53

Appendix D (cont'd)

	Earth Orbit	Planetary	Lunar
LF_2	474	790	338
LH_2	49	69	38
FLOX	474	790	338
CH_4	540	1050	360

The boiloff losses can be calculated by knowing the heat of vaporization, h_v, of the liquid. For this analysis, h_v is assumed a constant with pressure and temperature, and boiloff penalties are then:

$\Delta W_{B.O.} = \frac{Q}{h_v}$ lbs. These boiloff losses are:

Propellant Boiloff Losses, lbs. (100 psia venting; 68 psia for LH_2)

TABLE D6

	LO_2	LH_2	LF_2	LH_2	FLOX	CH_4
Earth Orbit	1580	4170	1820	2880	1980	215
Planetary	285	3850	520	2680	565	-0-
Lunar	3380	5500	3900	3760	4250	620

As can readily be seen, these boiloff losses are large and this presents the prospect of subcooling or active refrigeration to reduce these penalties. Subcooling the propellants allows one to take advantage of the thermal capacitance of the bulk fluid since the heat input causes the temperature of the bulk to rise rather than to vaporize propellants. The amount of heat which can be absorbed due to subcooling is $Q/m = C_p \Delta T$; where C_p is the specific heat of the propellant and ΔT is the temperature change. The amount of heat absorbed by the propellants to raise the bulk temperature to the boiling point after subcooling is shown below. The resulting vapor pressures for the various missions to be completed without venting are also shown. It can be seen that the planetary and earth orbital missions can be accomplished without venting and with relatively low pressures. The lunar mission, however, requires venting and the storage time one can achieve without venting versus the vapor pressure are shown.

Appendix D (cont'd)

Vapor Pressure for Non-Venting with Subcooling

TABLE D7

	$C_p\Delta T$,BTU/lb	Earth Orbit	Planetary	Lunar
LO_2	14.2	55 psia	35	Venting Req'd
LF_2	10.4	73	46	" "
FLOX	10.8	68	43	" "
CH_4	19.5	78	34	180

TABLE D8

Lunar Mission Storage Time (no venting), Days

Maximum Vapor Pressure	LO_2	LF_2	FLOX	CH_4
40 psia	464	422	422	337
60	551	493	493	416
80	614	550	550	486
100	653	593	593	530

From these results, it can be seen that with subcooling, planetary and earth orbital missions can be accomplished without venting. Lunar storability time can be extended to about 1/1/2 to 2 years without venting. The propellant that would be lost in boiloff for the full two year mission is

TABLE D9

Propellant Boiloff Losses with Subcooling (2 years on lunar surface)

LO_2	615 lbs
LF_2	1,350
FLOX	1,440
CH_4	337

To assess the boiloff weight penalties, the increase in tank size, support structure, and meteoroid shielding due to the increase in loaded propellant must be determined. The tank weights are proportional to the tank volumes and hence

$$\Delta W_{tank_{B.O.}} = \frac{W_{B.O.}}{W_P} W_{tank}$$

Appendix D (cont'd)

where $\Delta W_{tank_{B.O.}}$ = increased tank weight, lbs

$W_{B.O.}$ = weight of boiled off propellant, lbs

W_P = weight of useful propellant, lbs

W_{tank} = calculated tank weight to store useful propellant, lbs

Since the stage length will also be increased due to larger tanks, the meteoroid shield weight is assumed to increase approximately in proportion to the cube root of the volume change, or

$$\Delta W_{M.S.} = \frac{\Delta V_T^{1/3}}{V_o} W_{M.S.}$$

where $\Delta W_{M.S.}$ = increase in meteoroid shield weight, lbs

ΔV_T = increase in tank volume, ft^3

V_o = initial tank volume, ft^3

$W_{M.S.}$ = initial meteoroid shield weight, lbs

The boiloff weight penalties are

TABLE D10

	No Subcooling			Subcooling		
	ΔW_{tank}	$\Delta W_{M.S.}$	ΔW_{total}	ΔW_{tank}	$\Delta W_{M.S.}$	ΔW_{total}
LO_2	48	93	141	9	---	9
LH_2	973	654	1,627	876	561	1,437
			1,768			1,446
LF_2	57	72	129	20	---	20
LH_2	678	549	1,227	625	485	1,110
			1,356			1,130
FLOX	57	56	113	19	---	19
CH_4	21	47	68	11	---	11
			181			30

Appendix D (Con'd)

It can be seen that subcooling will substantially reduce the boiloff weight penalties for the lunar mission. Savings amount to over 320 pounds for a LO_2/LH_2 system, 230 pounds for LF_2/LH_2, and about 150 pounds for $FLOX/CH_4$. The problems associated with subcooling would have to be weighed against the saving that can be obtained to see if indeed it is desired.

It should be realized that these weight penalties for boiloff are only true penalties if boiloff occurs. If post injection planetary abort is necessary, the additional propellant not yet boiled off is usable. The engines must be operated fuel rich, however, to utilize these propellants since much more fuel will boil off then oxidizer. The resulting decrease in specific impulse will be more than offset by the increase in total impulse.

A preliminary investigation of active refrigerator systems indicates them to be feasible. Power requirements are the major problem area. The required refrigeration in watts electrical power is shown in Table D11. From reference 12, the efficiencies of presently available refrigerators as a function of load and operating temperature are estimated and also shown. For liquid hydrogen systems, this efficiency is around 0.4 percent, while it is about 2.5 percent for all the oxidizers and 4.0 percent for CH_4. The required electrical power necessary to operate these refrigerators continuously is also shown in Table D11. For LO_2, LF_2, FLOX, and CH_4, the required power is less than 400 watts, and could be supplied from the spacecraft power source which has a peak capacity of from 3 to 5 kilowatts. The LH_2 tanks, however, require from 3 to 5 kilowatts continuous power and this is excessive. Sizing power plants based on 600 pounds per kilowatt, a total refrigerator system weight can be estimated. Reference 12 presents data on refrigerator and radiator weights. The total refrigerator system weights are seen to be around 500 pounds for LO_2, LH_2, and FLOX, about 300 pounds for CH_4, and from 2200 to 3300 pounds for LH_2. These weights allow large savings in total stage gross weight at earth launch when compared to boiloff propellant and tank penalties.

Appendix D (cont'd)

A novel refrigeration concept discussed in Reference 12 is the Vuilleumier system. This system utilizes heat instead of electrical energy as the driving force. Only a small amount of electric power is required. If an isotope heat source is available, this system may have much merit for this application. No Vuilleumier machines have yet been built, however.

TABLE D11

REFRIGERATION SYSTEMS SUMMARY

	LO_2	LH_2	LF_2	LH_2	FLOX	CH_4
Heat Load, Maximum BTU/hr.	33.4	66.4	30.3	42.5	32.7	14.8
Watts	9.8	19.4	8.9	12.5	9.6	4.3
Power Efficiency, %	2.5	0.4	2.5	0.4	2.5	4.0
Required Power, kw	.39	4.85	.36	3.13	.38	.11
System Weight, lbs	519	3355	496	2223	512	274

A summary of the storability weight penalties (tankage, associated structure, and refrigerator weights) is shown below. The four cases shown are:

1. All propellants are loaded at launch with saturated vapor.
2. All propellants subcooled at launch.
3. All propellants except LH_2 subcooled at launch, LH_2 tanks refrigerated.
4. All propellant tanks refrigerated.

TABLE D12

Propellant Storability Weight Penalties, lbs.

	LO_2/LH_2	LF_2/LH_2	$FLOX/CH_4$
Case 1	1768	1356	181
2	1636	1247	30
3	3364	2243	30
4	3874	2719	786

Appendix D (cont'd)

The lightest system after boiloff results from subcooling all propellants. The lightest systems at earth launch would be the refrigerated systems. Refrigerated stages, however have this large thermal weight penalty throughout the mission duration.

Tank Design Pressures

The maximum tank operating pressure and hence the tank design pressures are set by the maximum vapor pressure. With refrigerated tanks, the vapor pressure can be held at a minimum. The design conditions are such that both earth orbital and planetary missions can be completed without venting or active refrigeration. The LH_2 tank will be designed for 68 psia since this is the vapor pressure that will result when the liquid has expanded to almost completely fill the ullage volume. For the other propellants, the design pressure is:

$$P_{tdes} = (P_o + \Delta P_{Heat} + NPSP) \times S.F.$$

where

P_o = Initial tank pressure = 15 psia

ΔP_{Heat} = Pressure rise due to heat input, psia

NPSP = Net positive suction pressure required by engine turbopumps = 5 psia

S.F. = Stress safety factor = 1.5

TABLE D13

Tank Design Pressures, psia

	P_o	ΔP_{Heat}	NPSP	P_F	P_{Des}
LH_2	15.0	50.0	5.0	70.0	105.0
LO_2	15.0	40.0	5.0	60.0	90.0
LF_2	15.0	58.0	5.0	78.0	117.0
FLOX	15.0	50.0	5.0	70.0	105.0
CH_4	15.0	63.0	5.0	83.0	124.0
Comp.A	15.0	-0-	5.0	20.0	30.0
MHF-5	15.0	-0-	5.0	20.0	30.0

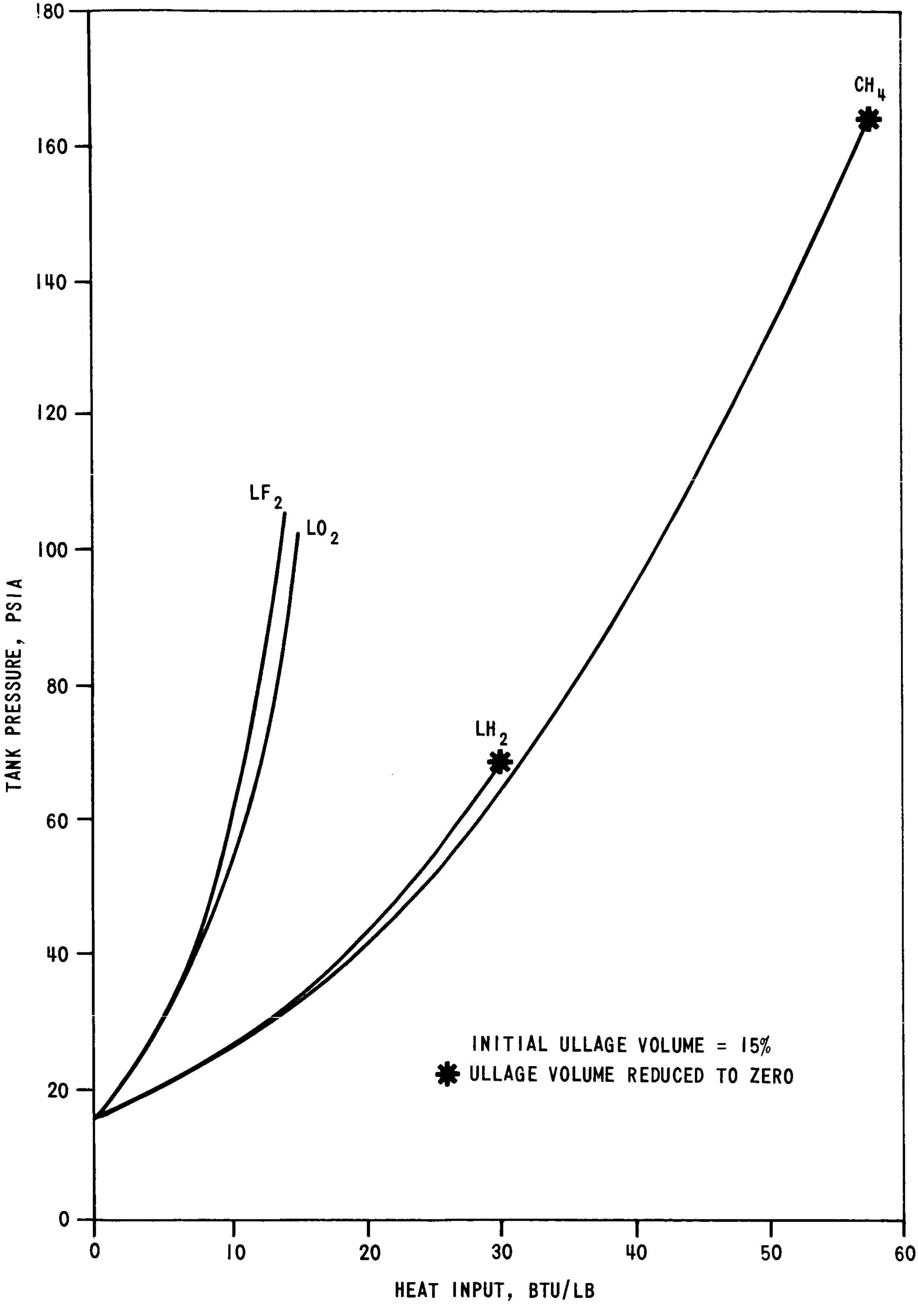

FIGURE DI - TANK PRESSURE vs. HEAT INPUT

APPENDIX E

STRUCTURES AND SUBSYSTEMS

No attempt was made at this time to go into detailed structural or subsystem designs. The weights of most of the items shown in Table E1 were estimated by pro-rating from other propulsion stage designs, references 1, 2, and 8. The pressurization system weights were calculated using a Tridyne* system for LO_2, LH_2, and CH_4, and using heated helium for the other propellants. The engine weight was estimated by assuming a conservative thrust to weight ratio of 75 for a thrust level of 30,000 pounds. This is for a single engine stage design. The effect of multiple engine designs is shown in Appendix F.

TABLE E1

Structural and Subsystem Weights, Lbs

	LO_2/LH_2	LF_2/LH_2	$FLOX/CH_4$	Compound A/MHF-5
Tank Supports	14 lbs	13	15	18
Support Trusses	25	22	25	28
Fittings	11	11	11	11
Propellant Orientation	22	22	22	22
Thrust Structure	270	242	265	300
Fairings	45	45	45	45
Base Heat Protection	45	45	45	45
Engine	400	400	400	400
Feed, Fill, & Drain Systems	120	111	130	167
Pressurization System	180	98	47	170

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* See next page

Appendix E (Cont'd)

	LO_2/LH_2	LF_2/LH_2	$FLOX/CH_4$	Compound A/MHF-5
RCS	100	100	100	100
Engine Gimble	40	40	40	40
Instrumentation & Equipment	500	500	500	500
	1772	1649	1645	1846

*Tridyne is a pressurization system developed at Rocketdyne which uses a small amount of catalytically ignited O_2/H_2 as a source for heating helium, the main pressurant gas.

APPENDIX F

STAGE CONFIGURATION

PM-II stage configurations employing one, two, and three engines are shown in Figures 2, 3, and 4. The single engine configuration is the simplest, shortest, and lightest design, but provides no engine-out capability. It is also relatively inefficient in the use of the available volume. It would use a torus oxidizer tank with a circular cross-section and an ellipsoidal fuel tank for minimum stage length.

The two engine design allows much more efficient use of the available space and results in a stage length about 2.5 to 4.0 feet shorter than either the one or three engine designs. This is possible if ellipsoid cross-sectional torus oxidizer tanks are used to enclose a cylindrical fuel tank with ellipsoidal domes. The two engine design provides some engine-out capability but an associated difficult control problem since the resultant thrust vector would be greatly misaligned from the center of gravity.

The three engine configuration provides engine out capability. If the engines are installed in line, the center engine could fail and the outer engines still adequately complete the missions. This, of course, necessitates sizing the center engine to complete all missions. The same must be true of the two outboard engines combined. For economy, all engines should be identical. Once again, relatively inefficient use is made of the available volume. This stage would be the longest and heaviest of the three but provide the most versatility and reliability. This stage would be about 1.5 feet longer than the single engine stage.

APPENDIX G

PROPULSION SYSTEMS

The engine performance and weights were based on the estimates of the major propulsion contractors and are consistant with those used in current contractural studies[24]. The specific impulse of the engines are slightly conservative since 460 seconds has already been demonstrated for LO_2/LH_2, better than 400 seconds is estimated for FLOX/CH_4, better than 470 for LF_2/LH_2, and around 350 to 360 for Compound A/MHF-5. It was intended, when possible, to keep the study conservative.

Engine thrust to weight ratios estimated by Pratt and Whitney Aircraft show 46 for an 8000 pound thrust engine and 65 for a 15,000 pound thrust engine. The engine thrust level must be large enough to satisfy the thrust to weight requirement for lunar ascent and return (minimum $F/W \approx 0.2$) and must be large enough such that the engine run time does not exceed approximately 5 minutes for planetary abort.

The gross weight at lift off from the lunar surface will be about 40,000 pounds. The minimum thrust level will therefore be 8000 pounds. The engine burn time is $t_b = \frac{W_P}{\dot{W}} = \frac{W_P I_S}{F}$

The burn time for the candidate propellants at various thrust levels is:

TABLE G1

Engine Burn Time, minutes

	LF_2/LH_2	LO_2/LH_2	FLOX/CH_4	Compound A/MHF-5
F = 8000 lbs	17.9	19.5	18.3	19.8
10,000	14.3	15.6	14.7	15.8
15,000	9.5	10.4	9.8	10.5
20,000	7.2	7.8	7.3	7.9
30,000	4.8	5.2	4.9	5.3
50,000	2.9	3.1	2.9	3.2

From these results, it can be seen that the single engine design should have a thrust of about 30,000 pounds. The major penalty of a higher thrust engine would be more weight.

Appendix G (Cont'd)

The two engine design would require thrust levels of 15,000 pounds each. The three engine configuration would require a thrust of 10,000 pounds. Using Pratt and Whitney weights and assuming engine weights are independant of propellants, the resulting engine weights for the stage configurations are:

TABLE G2

Engine Weights, lbs.

	Thrust/Engine	Weight/Engine	Total Engine Weight
One Engine	30,000 lbs	400 lbs	400 lbs
Two Engines	15,000	230	460
Three Engines	10,000	200	600

Appendix H - Life Support and Environmental Control Stage Addition

The Apollo and AAP earth entry modules are capable of sustaining life independently for less than 24 hours. For some of the contemplated missions with PM-II's, a lifetime of up to 10 days may be required of the EEM for a four man crew. A small stage addition can be attached to the EEM to provide the necessary functions to extend the EEM life time. From Reference 17, estimations were made of the required services, their weights, and the volume required to package them. These estimations are shown below:

TABLE H1

EPS	250 lbs
Fluid Storage	250 lbs
EC/LSS	265 lbs
Crew Provisions	45 lbs
Food	74 lbs
Water	55 lbs
High-Pressure O_2	50 lbs
Cryogenic O_2	550 lbs
Hydrogen	61 lbs
Personal Hygiene/Waste Management	36 lbs
Meteoroid Shielding and Structure	87 lbs
	1723 lbs

TABLE H2

Volume Requirements

Hydrogen	55 ft^3
Oxygen	35 ft^3
EPS	14 ft^3
All Other Equipment	20 ft^3
	124 ft^3

Appendix I - Propellant Physical Properties

Some of the important physical properties of the propellants considered are shown below:

TABLE I1

	LH_2	LO_2	LF_2	FLOX	CH_4	Comp A	MHF-5
Density, NBP, lb/ft^3	4.4	71.3	94.4	90.0	26.5	115.0	59.2
Boiling Point, °R	37	163	153	154	201	468	667
Freezing Point, °R	25	98	95	95	164	300	389
Heat of Vaporization, BTU/lb	195.3	91.6	71.5	74.8	219.8	.16	.8
Heat of Fusion, BTU/lb	25.2	5.9	5.8	5.8	26.1		
Specific Heat, BTU lb °R	3.39	.218	.180	.184	.528		.66